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APPLICATION OF CFD TECHNIQUES TOWARD THE VALIDATION OF NONLINEAR AERODYNAMIC MODELS

by

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SUMMARY

Applications of CFD methods to determine the regimes of applicability of nonlinear models describing the unsteady aerodynamic responses to aircraft flight motions are described The potential advantages of computational methods over experimental methods are discussed and the concepts underlying mathematical modeling are reviewed The economic and conceptual advantages of the modeling procedure over coupled, simultaneous solutions of the gasdynamic equations and the vehicle's kinematic equations of motion are discussed The modeling approach, when valid, eliminates the need for costly repetitive computation of flowfield solutions For the test cases considered, the aerodynamic modeling approach is shown to be valid

1 INTRODUCTION

Predictions of aircraft flight motions, design of aircraft control systems, and development of realistic piloted flight simulators all hinge on accurate knowledge of the unsteady aerodynamic forces and moments acting on the maneuvering aircraft Prediction of these unsteady airloads is complicated by the fact that the instantaneous flowfield surrounding the aircraft is not determined solely by the instantaneous values of the motion variables such as angle of attack, pitch rate, and control deflection angles In general, the instantaneous state of the flowfield, and thus the loading, depends on the history of the motion, that is, on all states experienced by the flowfield during the maneuver prior to the instant in question Time-history effects are accentuated by requirements for aircraft agility, which lead to flight in the high-angle-of-attack regime and rapid changes in the orientation of the aircraft As a result, highly-maneuverable aircraft can experience nonlinear and unsteady airloads due to large regions of three-dimensional separated flow, concentrated vortex flows, and the presence and movement of shock waves

Complete evaluation of time-history effects in wind-tunnel experiments would require the availability of an apparatus capable of simulating, at suitable rates, the complete range of motions the aircraft would experience in flight, and capable of measuring the aerodynamic response to those motions By coupling this apparatus with a computer, the aircraft's equations of motion could be integrated in time from specified initial conditions The computed aircraft flight attitude and angular rates would determine the new position of the apparatus, while the measured aerodynamic forces and moments would be input to update the computed aircraft motion Although, in principle, capable of accounting for all time-history effects, such a general wind-tunnel apparatus is, unfortunately, far more easily envisioned than constructed As a result the conventional method of predicting aircraft flight motions relies on an alternative approach, that of mathematical modeling

In the modeling approach, one attempts to specify a *form* for the aerodynamic response that remains the same in the determination of the aerodynamic response to all motions of interest Ideally, a mathematical model enables one to synthesize the response to a general maneuver from the known aerodynamic responses of the aircraft to a limited number of specific, *characteristic* motions Wind-tunnel evaluations of the responses to the characteristic motions are, in principle, more easily accomplished than simulations of the general maneuvers The development of mathematical models capable of accurately describing the variety of aerodynamic phenomena acting on aircraft maneuvering in the nonlinear flight regime has been a subject of ongoing interest (Refs 1,2), and continues to be addressed in this symposium (Refs 3,4)

The utility of a proposed mathematical model is dependent upon the range of vehicle motions and aerodynamic phenomena which it can encompass. Determining the range of validity of a candidate mathematical model would require 1) determining the aerodynamic response of an aircraft to the set of characteristic motions called for by the model, 2) predicting a wide variety of aircraft motion histories using the model and the determined responses, and 3) comparing the predicted motion histories against actual flight histories The validity of the model would be demonstrated by a match between the predicted and flight motion histories While such a model validation procedure is straightforward in principle, it is difficult to accomplish in practice Flight-test data are expensive to obtain, and accurate determination of motion histories during extreme maneuvers is difficult (see Ref 5 for a discussion of flight-test difficulties) Further, accurate wind-tunnel determination of the unsteady responses to the characteristic motions is itself nontrivial (see Refs 6,7 for a description of relevant experimental techniques) In addition, in the comparison of motion histories, small changes in the values of the terms of the model may integrate to produce large deviations between the actual and predicted motions Thus, few, if any, satisfactory model validations have been accomplished for a general set of flight maneuvers

Fortunately, the remarkable advances which have occurred in computer technology and computational fluid dynamics now offer the aerodynamicist a promising alternative approach Today, we can envision utilizing CFD techniques to study unsteady three-dimensional flows, both to investigate aerodynamic time-history effects and to validate aerodynamic mathematical models

A straightforward computational approach to investigating time-history effects (assuming the availability of codes and computer resources adequate to solve suitably modeled equations of turbulent flow around an aircraft) would be to solve the flowfield equations *simultaneously* with the aircraft's equations of motion Such a coupled approach is shown schematically in Fig 1a This is, of course, the CFD analog of the general wind-tunnel apparatus described above, with computation of the instantaneous flowfield replacing the need for measuring the aerodynamic response By using CFD techniques one avoids the experimental difficulties which hinder wind-tunnel measurements of unsteady flows Results of these coupled computations would be complete time histories of the aerodynamic response and of the vehicle motion It is noteworthy that computations involving the coupled-equations approach have recently been carried out for several unsteady two-dimensional inviscid flows (Refs 8 - 10), and for at least two unsteady two-dimensional viscous flows (Refs 11,12) However, lack of computational resources has, to date, precluded undertaking the analogous computations for (even steady) three-dimensional viscous flows

Although coupling of the flowfield equations and the aircraft's inertial equations of motion is, in principle, an exact approach to accounting for aerodynamic time-history effects in predicting the response to arbitrary maneuvers, it inevitably will be a very costly one This will be particularly true for maneuvers at high incidence, where the airloads depend nonlinearly on the motion variables Under such conditions the aircraft can experience widely varying motion histories, even if they are started from closely-spaced initial conditions Thus, to completely evaluate an aircraft's performance envelope, a large number of computational cases, each involving the coupled equations, would be required to cover all possible sets of initial conditions Since the motion and the aerodynamic response are inextricably linked in the coupled approach, the flowfields must be recomputed for each change in initial conditions

In contrast to the coupled-equations approach, utilizing mathematical modeling in conjunction with CFD techniques can eliminate the need for repetitive computation of flowfield solutions, and will lead to more efficient use of computational resources The modeling approach is shown schematically in Fig 1b As has been previously mentioned, in formulating a model one attempts to identify a set of characteristic motions from whose aerodynamic response one can generate the response to general maneuvers Within the regime of validity of the mathematical model, computational evaluation of the aerodynamic terms specified by the model would be required only once, whereupon they could be utilized over a range of motion variables and flight conditions. Flight motions could then be predicted by solving the aircraft's equations of motion independently of the flowfield computations, at far less cost

Presupposing the availability of codes adequate to solve the equations governing turbulent flows around an aircraft, evaluation of the validity of a candidate aerodynamic mathematical model can itself be easily accomplished with the use of CFD techniques Two validation procedures can be envisioned The first method parallels the experimental validation procedure discussed above, that is, by comparing vehicle motion histories To carry out the validation, the aerodynamic responses to the characteristic motions would be evaluated from flowfield computations Once obtained, these responses can be used, together with the model, to predict a series of motion histories On the other hand, by using the coupled-equations approach, motion histories which take exact account of time-history effects can be obtained for the same set of initial conditions Both series of motion histories are subject to precisely the same set of approximations made in computing the flowfield Differences, if any, between the motion histories must be attributable to the assumptions made in the modeling process Thus, the validity of the mathematical model would be demonstrated by agreement between the motion histories The second computational model validation procedure would involve computing the aerodynamic responses to the characteristic motions, and using the mathematical model and the computed responses to obtain the aerodynamic (rather than the motion) response to a complex, but specified, motion The response to the complex motion would also be computed directly, and compared to the one obtained using the mathematical model Here the validity of the model would be demonstrated by agreement between the two aerodynamic response histories

In this paper we review two mathematical model validation efforts effected with CFD techniques In the first, Chyu and Schiff (Ref 10) utilized two-dimensional unsteady-flow computations to validate a nonlinear aerodynamic mathematical model for the case of a freely-deflecting flap hinged to a stationary airfoil and immersed in a transonic flow Validation of the model was demonstrated by close agreement of flap motion histories obtained with the model and those obtained with the coupled-equations approach More recently, Katz and Schiff (Ref 13) investigated the validity of a nonlinear aerodynamic mathematical model for low-rate multi-degree-of-freedom motions of a delta wing maneuvering at large angles of attack. Although computation of three-dimensional unsteady viscous flows by means of modeled equations of turbulent flow is not yet feasible, computation of unsteady three-dimensional inviscid flows by means of potential equations is currently possible The authors of Ref 13 employed a nonlinear vortex-lattice method to compute the aerodynamic responses to the characteristic motions called for by the model Aerodynamic responses to complex specified motions were generated by means of the model and also from direct flowfield computations The validity of the mathematical modeling concepts for the delta wing maneuvering in the high-angle-of-attack regime was demonstrated by close agreement of the corresponding force histories In the following sections we discuss the procedures and results of Refs 10 and 13, focusing in particular on the validation of the mathematical models and on the costs of the modeling approach relative to those of the coupled-equations approach

2. AERODYNAMIC MATHEMATICAL MODELS

The form that an aerodynamic mathematical model takes depends upon the coordinates used and variables used to describe the motion, and on the types of aerodynamic phenomena that the model can encompass In a series of papers, Tobak and his colleagues used the concept of a nonlinear indicial

function to derive aerodynamic models to describe, within certain assumptions, nonplanar maneuvers of bodies of revolution (Ref 14) and of aircraft (Refs 15,16) In this section an alternative derivation is reviewed which puts clearly into perspective the assumptions made in deriving the models, and the resulting limitations of the resulting models to accommodate certain types of aerodynamic phenomena

2 1 Model Describing Aircraft Maneuvers

Consider a maneuvering aircraft whose center of mass travels an essentially straight-line path at constant flight velocity The maneuvers can be described in terms of an aerodynamic axis system, in which the orientation of the aircraft relative to the oncoming wind is specified in terms of pitch and roll coordinates (Fig 2) In this axis system the resultant angle of attack, σ, is defined as the angle between the body-fixed longitudinal X axis and the flight velocity vector The plane containing σ is called the resultant-angle-of-attack plane The roll, or bank, angle ψ is the angle between the normal to the resultant-angle-of-attack plane and a body-fixed axis (normal to the X axis) which lies in the plane of the wing The resultant-angle-of-attack plane is free to rotate about the velocity vector with a rotation rate, measured relative to an inertia frame, of ϕ Knowledge of the history of the maneuver is equivalent to knowledge of the histories of σ, ψ, and ϕ

The aircraft undergoes a maneuver beginning at time zero The instantaneous aerodynamic response to the maneuver at time t depends, in general, on the entire history of the maneuver That is, the response depends on the histories of the motion variables over the interval from 0 to t Mathematically, the response is a *functional* of the motion variables Thus the pitching-moment coefficient, for example, can be expressed in the square-bracket notation of Volterra, as

$$C_m(t) = C_m[\sigma(\xi), \psi(\xi), \phi(\xi)] \tag{1}$$

where ξ is a dummy time variable ranging from 0 to t Note that $\phi(\xi)$ appears in Eq (1), rather than $\phi(\xi)$, because the aerodynamic response is independent of the orientation of the angle-of-attack plane (in contrast to that of the pilot) and depends only on the rate at which the σ plane is rotating

If the motion is assumed to be differentiable, one can expand each of the variables $\sigma(\xi)$, $\psi(\xi)$, and $\phi(\xi)$ in Taylor series about ξ = t Knowledge of the past history of the motion is equivalent to knowing all of the coefficients of the series Substituting these coefficients for the motion histories in Eq (1) converts the pitching-moment response from a functional to a function, albeit a function of an infinite number of variables Thus, the pitching-moment coefficient becomes

$$C_m(t) = C_m(\sigma(t), \sigma(t), \ddot{\sigma}(t), \ldots, \psi(t), \psi(t), \ddot{\psi}(t), \ldots, \phi(t), \ddot{\phi}(t), \ldots) \tag{2}$$

The assumption that the motion history is differentiable immediately restricts the model to cases where the aerodynamic forces and moments are continuous If sudden jumps in the aerodynamic response occurred, the aircraft would experience discontinuous accelerations, and this would violate the assumption of a differentiable motion history Restriction to continuous forces rules out the capacity of the resulting mathematical model to treat the bi-valued aerodynamic responses characterizing static hysteresis and both time-invariant and time-dependent subcritical bifurcations (cf Ref 3) Within the restriction to continuous, single-valued, aerodynamic responses, one can make a further assumption that, for the slowly-varying motions typifying aircraft rigid-body motions, the aerodynamic response will have only a negligible dependence on the higher-order rate terms such as $\ddot{\sigma}(t), \sigma(t), \ddot{\psi}(t), \psi(t), \ddot{\phi}(t)$, etc , and thus they may be neglected in Eq (2) This reduces the pitching-moment response to a function of a finite, rather than an infinite, number of variables Consistent with the assumption of slow motion, one is justified in expanding Eq (2) in a Taylor series about $\sigma = 0$, $\psi = 0$, and $\phi = 0$ The resulting model is

$$\begin{aligned} C_m(t) = & C_m(\sigma(t), \psi(t), 0,0,0) + \frac{\psi b}{2U} C_{m_{\dot\psi}}(\sigma(t), \psi(t), 0,0,0) \\ & + \frac{\sigma b}{2U} C_{m_{\dot\sigma}}(\sigma(t), \psi(t), 0,0,0) + \frac{\phi b}{2U} C_{m_{\dot\phi}}(\sigma(t), \psi(t), 0,0,0) \end{aligned} \tag{3}$$

where b is the wing span, and the zeros refer to the motion rates Equivalent expressions for the yawing- and rolling-moment coefficients, C_n and C_l, and for the axial-, side-, and normal-force coefficients, C_X, C_Y, C_N, are obtained by substituting these coefficients wherever C_m appears in Eq (3) Again, the model applies to slowly varying motions of the aircraft, although the values of σ and ψ may be large The mathematical model is seen to contain four terms, and each term can be identified with a specific, characteristic motion from which it may be evaluated Thus, $C_m(\sigma(t), \psi(t), 0,0,0)$ is the pitching-moment coefficient that would be evaluated in a steady flow with σ and ψ held fixed at $\sigma(t)$, $\psi(t)$ The second term, $C_{m_{\dot\psi}}$, is the contribution to the pitching-moment coefficient due to rolling motion, and can be evaluated for small-amplitude oscillations in ψ about ψ = const, with σ held fixed and ϕ fixed at zero Similarly, the term C_{m_σ} is the contribution to the pitching-moment coefficient due to pitching motions, and can be evaluated for small-amplitude planar oscillations in σ about σ = const, with ψ held fixed and ϕ fixed at zero. The last term, $C_{m_{\dot\phi}}$, is the rate of change of the pitching-moment coefficient with coning-rate parameter, $\phi b/2U$, evaluated at ϕ = 0, that would be determined from a steady coning motion with σ = const, ψ = const, and ϕ = const These characteristic motions are illustrated in Fig 2

2 2 Model Describing Flap Motions

Derivation of an aerodynamic mathematical model applicable to slowly-varying motions of a flap on an airfoil parallels that of the model, Eq (3), describing the response to aircraft maneuvers For the airfoil flap motions considered in Ref 10, the coordinates illustrated in Fig 3 were employed The airfoil was a NACA 64A010 section hinged at the 75% chord point The forward portion of the airfoil was held fixed parallel to the oncoming stream, while the flap was free to move about its pivot point Positive values of the flap deflection angle, σ_f, and of the flap hinge-moment coefficient, C_h, are shown in Fig 3 As illustrated, a positive value of C_h would tend to increase the flap deflection angle For the single-degree-of-freedom flap motion, the hinge-moment coefficient is, in general, a functional of the flap deflection angle history, i e , $C_h(t) = C_h[\sigma_f(\xi)]$ By means of the same logic described above for aircraft maneuvers, the functional can be reduced to the form

$$C_h(t) = C_h(\sigma_f(t),0) + \frac{\dot{\sigma}_f l}{V} C_{h_{\dot{\sigma}_f}}(\sigma_f(t),0) \tag{4}$$

where the reference length l is the chord length. As before, the model applies to slowly-varying motions of the flap, although the values of the flap deflection angle can be large Again, each of the terms in Eq (4) is identified with a particular characteristic motion from which it may be evaluated Thus, the term $C_h(\sigma_f(t),0)$ is the hinge-moment coefficient that would be evaluated in a steady flow with the flap deflection angle held fixed at $\sigma_f(t)$ The remaining term, $C_{h_{\dot{\sigma}_f}}$, is the contribution to the hinge-moment coefficient due to flap motions, and can be evaluated for small-amplitude oscillations of the flap about a mean value of σ_f held fixed at the instantaneous value of $\sigma_f(t)$

2 3 Issues of Model Applicability

Consistent with the assumptions made in their development, aerodynamic mathematical models at the level of Eqs (3) and (4) are subject to the following restrictions 1) the response to a steady motion is itself steady, 2) the response is a single-valued (although allowably nonlinear) function of the orientation of the body, and 3) the responses are linear in the motion rates Restriction 1 rules out the possibility of modeling time-dependent aerodynamic bifurcations, that is, development of time-varying (periodic, quasi-periodic, or chaotic) responses to a steady motion, e g , the time-varying flow observed surrounding a stationary circular cylinder in crossflow at Re $\geq$ 50 However, recent modeling efforts (see Ref 3), based on the concept of Fréchet differentiability of the aerodynamic response, include treatment of such time-varying flows Restriction 2 precludes modeling the bi-valued aerodynamic responses characterizing static aerodynamic hysteresis which have been observed in cases of vortex asymmetry on slender bodies of revolution and on slender delta wings However, this behavior can also be accomodated (see Ref 1, and more recently, Ref 3) The third restriction precludes modeling the nonlinear dependence of the aerodynamic response on the motion rates, and in particular, nonlinear dependence on $\dot{\phi}$, the coning rate Such nonlinear variation with $\dot{\phi}$ has been observed experimentally (see Ref 17), and can be incorporated in the model by performing the Taylor series expansion about $\sigma = 0$, $\psi = 0$, and $\dot{\phi} = \dot{\phi}(t)$ rather than about $\sigma = \psi = \dot{\phi} = 0$, as was done to obtain Eq (3) This would result in an aerodynamic model equivalent to the one derived previously (Ref 16) using the nonlinear indicial function approach

The utility of the aerodynamic modeling approach depends on the ability of the model to treat the aerodynamic phenomena which occur in flight In applying models analogous to Eq (3), the general aircraft motion is decomposed into a sum of characteristic motions The aerodynamic response to the general motion is *modeled* as a sum of responses to the characteristic motions The actual response to the general motion will differ from the modeled response if aerodynamic phenomena excluded in developing the model are present The assumption made in developing Eq (3), that the aerodynamic responses are continuous, single-valued functions of the motion variables, restricts the model to cases where neither hysteresis nor time-dependent aerodynamic bifurcations occur Within this restriction, the remaining causes for failure of the model to predict a general response would be either 1) significant nonlinear dependence of the aerodynamic responses on rates of motion within the range of rates actually experienced in flight, or 2) presence of significant interactions between responses to pairs characteristic motions Examples of such interactions include those between responses to pitch oscillations and coning motion or between responses to roll oscillations and coning motion, i e , terms such as $C_{m_{\dot{\sigma}\dot{\phi}}}$ or $C_{m_{\dot{\psi}\dot{\phi}}}$ which have been excluded in deriving Eq (3)

2 4 CFD Requirements for Validating Models

As discussed in Sec 1, validation of a candidate mathematical model with the use of CFD methods involves computation of both the aerodynamic response to characteristic motions and the response to either specified complex motions or coupled-equations responses From a CFD standpoint, the boundary conditions for the low-rate characteristic pitching, rolling, and coning motions are linear perturbations in σ, ψ, and $\dot{\phi}$, respectively, about the zero values associated with the steady characteristic motion If flowfields were governed by linear partial differential equations, the linear boundary conditions guarantee that the responses to the characteristic motions would be linear in the rates Also, the response to any general motion could be obtained from superposition of the responses to the characteristic motions, for such a solution would satisfy both the PDE's and BC's exactly In this circumstance, models at the level of Eq (3) would be exact for all cases

However, the actual equations governing fluid flow are nonlinear, and the presence of nonlinear convection terms in the momentum equations give rise to the possibility of all the aerodynamic phenomena discussed above Thus, to be useful in assessing the validity of a model, the computational method must be based on nonlinear flowfield equations Approximating the flowfield equations with linearized PDE's will clearly be unsatisfactory for validating nonlinear models Such models will

always appear to be valid, since the linearized flowfield equations cannot capture nonlinear aerodynamic effects Thus, validation of limited aerodynamic models analogous to those of Eqs (3) and (4) would require, at a minimum, a CFD method capable of demonstrating nonlinear aerodynamic behavior with the motion rates In Ref 10, Chyu and Schiff used a time-accurate method based on the nonlinear Euler equations to demonstrate the validity of the model, Eq (4), for the case of an oscillating flap The Euler equations are capable of capturing nonlinear behavior of both the steady and unsteady flowfield responses Similarly, in Ref 13, Katz and Schiff utilized a nonlinear vortex-lattice method to investigate the ability of the equivalent model, Eq (3), to describe the nonlinear aerodynamic response to low-rate maneuvers of a delta wing In the vortex-lattice technique, the flowfield is assumed to be governed by the incompressible potential equation, $\nabla^2\phi = 0$ Leading-edge separation is modeled by permitting vortex panels to shed from the leading edge of the wing, and the free vortex sheet is allowed to roll up in response to the local velocity fields Although the PDE which describes the flowfield is linear, the boundary conditions are not Allowing the free vortex sheets to roll up permits the method to encompass a nonlinear variation, if present, of the aerodynamic response with the rates (see Ref 18)

Methods based on nonlinear, inviscid flowfield equations would appear adequate to validate aerodynamic models analogous to those of Eqs (3) and (4) These computational methods cannot, however, incorporate aerodynamic phenomena such as onset of vortex asymmetry, vortex breakdown, and time-varying response to fixed boundary conditions Thus, validation of aerodynamic models which purport to include these phenomena will require the use of methods based on nonlinear viscous flowfield equations

3 VALIDATION FOR MANEUVERING DELTA WING

In the study reported in Ref 13, CFD solutions were carried out for the unsteady three-dimensional flowfield surrounding a sharp-leading-edge delta wing to demonstrate the validity of the multi-degree-of-freedom mathematical model, Eq (3) The flowfield was assumed to be governed by the potential equations, and a vortex-lattice method (VLM) was applied to solve the time-dependent equations

3 1 Model Validation Procedure

The particular wing considered in Ref 13 was a slender delta wing having an aspect ratio of unity (leading-edge sweep angle = 75 96°) The center of mass was fixed at the wing half-chord (x/c = 0 50) The values of the resultant angle of attack and roll angle considered are shown in Fig 4 The resultant angle of attack σ ranged from 20° to 30°, while the roll angle ψ ranged from 0° (wing level) to 10° The dimensionless rates $\dot{\sigma}b/2U$, $\dot{\psi}b/2U$, and $\dot{\phi}b/2U$, ranged up to 0 15 The procedure utilized for validating the aerodynamic mathematical model applicable to the wing had three phases These were

1) Evaluate the aerodynamic responses to the characteristic motions from vortex-lattice computations

2) Generate aerodynamic force and moment response histories to prescribed complex motions using the aerodynamic mathematical model and the aerodynamic data evaluated in phase 1

3) Compare the histories obtained in phase 2 with force and moment histories that are, in principle, exact within the framework of the computational technique, namely those obtained by directly the vortex-lattice method to compute the response to the identical complex motions

Demonstration of the validity of the mathematical modeling approach, as applied to the delta wing maneuvering in the high-angle-of attack regime, hinged on finding close agreement between the force and moment time-histories obtained from the two approaches

3 2 Numerical Technique

In the computational procedure, the surface of the wing is divided into a number of bound vortex panels The strengths of the bound panels are determined at each time step during the computation to enforce the boundary condition that there be no flow through the solid wing Information describing the wing's maneuver enters the computation through the solid-surface boundary condition Time-evolution of the wake behind the wing is modeled by allowing vortex panels to shed from the trailing edge at each time step These wake panels have fixed strength and, upon leaving the wing, move with the local fluid velocity

When a delta wing is maneuvering at high angle of attack, flow separates near the wing leading edges, and the separated fluid rolls up above and behind the wing to form concentrated vortices For a sharp-edged wing, the separation line is essentially fixed at the sharp leading edge, and does not vary with changes in Reynolds number In the vortex-lattice method, leading-edge separation is modeled in a manner analogous to that of trailing-edge separation, by allowing vortex panels to shed from lines of separation specified to remain at the leading edges and permitting them to move with the local flow velocity Specification of the bound vortex strengths and free vortex positions yields the pressure distribution on the wing and, in turn, the nonlinear unsteady airloads (cf. Refs 18-21). Details of the numerical method and a discussion of the accuracy of the computed results are found in Ref. 18

3 3 Aerodynamic Response to Characteristic Motions

To obtain the aerodynamic data required by the mathematical model, Eq (3), computations were carried out for the wing in each of the four characteristic motions shown in Fig 2 at each value of resultant angle of attack and roll angle shown in Fig 4 The steady-state term $C_k(\sigma(t), \psi(t))$, where C_k denotes any of the force or moment coefficients, was obtained from a computation in which the resultant angle of attack and roll angle were held fixed, and the flowfield was allowed to evolve until it reached a steady state In an analogous manner the term $C_{k_{\dot\phi}}(\sigma(t), \psi(t))$ was obtained from a series of computations for steady coning motion in which the resultant angle of attack, roll angle, and coning-rate parameter were fixed, and the flowfield was allowed to evolve to a steady state Note that to an observer fixed in the moving wing, the flowfield due to a steady coning motion is indeed time-invariant The coefficient was then determined from the observed rate of change of the moment with coning-rate parameter, $\partial C_k/\partial(\dot\phi b/2U)$, evaluated at $\dot\phi = 0$

The aerodynamic coefficient due to pitch oscillations, $C_{k_{\dot\sigma}}(\sigma(t), \psi(t))$, was evaluated from small-amplitude harmonic pitch oscillations about the mean values of resultant angle of attack and roll angle shown in Fig 4 The wing was specified to move according to

$$\begin{aligned} \sigma &= \sigma_0 + \sigma_1 \sin\omega_1 t \\ \psi &= \psi_0 \\ \phi &= 0 \end{aligned} \tag{5}$$

The amplitude of the harmonic motion, σ_1, was specified to be less than 2° The aerodynamic damping coefficient was evaluated from the component of the aerodynamic response that was 90° out of phase with the wing motion This rationale is easily seen by substituting the conditions describing the pitch oscillations (Eq (5)) into the aerodynamic model to obtain (after a Taylor-series expansion about $\sigma = \sigma_0$ and omission of terms of $O(\sigma_1^2)$)

$$C_k(t) = C_k(\sigma_0, \psi_0) + \sigma_1 \sin\omega_1 t \frac{\partial C_k(\sigma_0, \psi_0)}{\partial\sigma} + \frac{\omega_1 \sigma_1 b}{2U} \cos\omega_1 t C_{k_{\dot\sigma}}(\sigma_0, \psi_0) \tag{6}$$

The coefficients in Eq (6) were obtained from a Fourier integration of the response over one cycle of the motion, as shown for the normal-force coefficient response in Fig 5. Thus

$$C_k(\sigma_0, \psi_0) = \frac{1}{2\pi} \int_0^{2\pi} C_k(t) d(\omega_1 t) \tag{7}$$

$$C_{k_\sigma}(\sigma_0, \psi_0) = \frac{1}{\pi\sigma_1} \int_0^{2\pi} C_k(t) \sin\omega_1 t d(\omega_1 t) \tag{8}$$

$$C_{k_{\dot\sigma}}(\sigma_0, \psi_0) = \frac{2U}{\pi b \omega_1 \sigma_1} \int_0^{2\pi} C_k(t) \cos\omega_1 t d(\omega_1 t) \tag{9}$$

The steady-state coefficient and its slope can be obtained either from the computations of the oscillatory motion (Eqs (7) and (8)) or, preferably, from the computations of the steady motion described earlier

In an analogous manner, the coefficient due to roll oscillations, $C_{k_{\dot\psi}}(\sigma(t), \psi(t))$, was evaluated for small-amplitude harmonic roll motions where

$$\begin{aligned} \sigma &= \sigma_0 \\ \psi &= \psi_0 + \psi_1 \sin\omega_2 t \\ \phi &= 0 \end{aligned} \tag{10}$$

and the amplitude of the motion, ψ_1, was specified to be less than 2° The roll damping coefficient was obtained from

$$C_{k_{\dot\psi}}(\sigma_0, \psi_0) = \frac{2U}{\pi b \omega_2 \psi_1} \int_0^{2\pi} C_k(t) \cos\omega_2 t d(\omega_2 t) \tag{11}$$

The results of the computations for the characteristic motion are shown in Figs 6-8 Generation of these diagrams of the aerodynamic coefficients required 36 individual computations, one for each of the four characteristic motions at the nine combinations of resultant angle of attack and roll angle shown in Fig 4 The results confirm that, over the range of angles of attack and roll and over the range of rates considered, the aerodynamic responses to the characteristic motions are linear in the motion rates, and are single-valued functions of the angles Thus, barring the presence of significant nonlinear interactions between the responses to pairs of characteristic motions, the aerodynamic model, Eq (3), should prove to be valid

3 4 Aerodynamic Response to Prescribed Complex Motions

The prescribed complex motions combined pitch oscillations, roll oscillations, and coning motion The combined motions all had the basic form

$$
\begin{aligned}
\sigma &= \sigma_0 + \sigma_1 \sin \omega_1 t, & \dot{\sigma} &= \omega_1 \sigma_1 \cos \omega_1 t \\
\psi &= \psi_0 + \psi_1 \sin \omega_2 t, & \dot{\psi} &= \omega_2 \psi_1 \cos \omega_2 t \\
\phi &= \omega_3 t, & \dot{\phi} &= \omega_3
\end{aligned}
\tag{12}
$$

Aerodynamic response histories of the pitching-moment, rolling-moment, and normal-force coefficients were computed from Eq (3), with the aerodynamic coefficients obtained from table look-ups of the data shown in Figs 6-8, and values of σ, ψ, and ϕ obtained from Eq (12) The aerodynamic responses to the combined motions were also obtained from direct VLM computations As was mentioned earlier, the use of the identical vortex-lattice method to evaluate both the nonlinear responses to the characteristic motions and the responses to the general large-amplitude motions ensures a consistent treatment of the time-history effects Thus discrepancies, if present, between results obtained using the modeling approach and those obtained from the direct VLM computations must be attributed to the inadequacy of the aerodynamic model

3 5 Validation of the Mathematical Model

The motions considered in Figs 9 and 10 combine pitch and roll oscillations, having amplitudes of 3° and reduced frequencies ranging up to 0 15, superimposed on a steady coning motion in which $\sigma_0 = 25°$ and $\psi_0 = 5°$ Typical responses of the pitching-moment, rolling-moment, and normal-force coefficients to these motions are shown in Figs 9 and 10 as functions of wing-chord lengths of travel, Ut/c In each figure the dotted lines show the histories obtained from the mathematical model, Eq (3), while the solid lines indicate the results obtained from direct VLM computations For the direct computations, the overshoot indicated at the beginning of each time-history occurs because the motion was started impulsively from rest It will be recalled that the mathematical models described in Sec 2 were obtained under the assumption of slowly-varying motions, and are not expected to model the impulsive start If the short initial transient period is excluded, in both cases the aerodynamic responses obtained from the model show reasonable agreement with those obtained from direct computation The differences between the results obtained from the model and those from direct VLM computation can be attributed to errors in the interpolations in the table look-ups based on Figs 6-8 that were required to obtain the coefficients of the model This would indicate that, for these cases, no significant interactions existed between responses to pairs of characteristic motions, such as the interaction between the responses to the pitch oscillation and coning motion, or between the responses to the roll oscillation and coning motions

The agreement shown in Figs 9 and 10 would tend to indicate that, over the range of angles and motion rates considered in Ref 13, the mathematical model presented in Eq (3) is adequate to describe the aerodynamic response to complex motions of the delta wing In actuality, for the range of pitch and roll rates considered, the contributions of the pitch-damping and roll-damping terms in Eq (3) were almost negligible As a result, the aerodynamic interactions between pairs of the characteristic motions, which are of higher order than the damping terms, must be negligible In these circumstances it is not surprising that the mathematical model would appear to be validated Thus, while the VLM computations confirm the validity of the model for the cases considered, the cases themselves do not conclusively demonstrate the limits of the range of motions and rates for which the model is valid Nevertheless, the procedure discussed indicates the way in which CFD methods can be used to validate a candidate mathematical model

4 VALIDATION FOR FLAP MOTIONS

In the study reported in Ref 10, CFD solutions were obtained for the two-dimensional unsteady transonic flowfield surrounding a wing and moving flap (Fig 3) to demonstrate the validity of the nonlinear mathematical modeling concepts The flowfield was assumed to be governed by the time-dependent inviscid Euler equations, and an implicit time-accurate finite-difference technique was applied to obtain the solutions The flowfield equations were solved on a moving body-conforming computational mesh, which deformed in response to the flap motion (Fig 11) Details of the numerical technique, boundary conditions, and mesh-generation method are found in Ref 10

4 1 Model Validation Procedure

The airfoil considered in Ref 10 was a NACA 64A010 section, hinged at the 75% chord point The forward portion of the airfoil was held fixed parallel to the oncoming stream, while the flap was free to pivot about its hinge point with deflection angles ranging up to 20° The free-stream Mach number was held fixed at 0 8 Validation of the aerodynamic mathematical model involved

1) evaluating the aerodynamic response to the characteristic motions from Euler-equation computations

2) utilizing the equation governing mechanically unconstrained motions of the flap,

$$I\ddot{\sigma}_f(t) = qSlC_h(t) \tag{13}$$

together with the aerodynamic mathematical model, Eq (4), to predict a series of flap motion histories

3) simultaneously solving the flowfield equations and flap-motion equation to obtain "exact" flap motion histories with the same initial conditions

The validity of the modeling approach for this nonlinear, transonic flow case was demonstrated by close agreement of the motion histories obtained from the two approaches

4 2 Hinge-Moment Response to Characteristic Motions

To obtain the hinge-moment coefficients required by the mathematical model, computations were carried out for the flap in both required characteristic motions The steady-state term $C_h(\sigma_f)$ was obtained from a computation in which the flap deflection angle was held fixed A time-invariant mesh was employed, and the flowfield was allowed to evolve until it reached a steady state The resulting static hinge-moment coefficient, obtained from spatial integration of the surface-pressure distributions, is shown in Fig 12 The hinge moment is statically stabilizing, tending to oppose the flap deflection The nonlinear behavior of the static hinge moment is associated with rearward movement and increase in strength of the upper-surface shock wave with increasing flap deflection For low flap deflection angles, at the Mach number considered, the shock wave was located ahead of the flap hinge point As the flap deflection was increased beyond 3°, the shock moved onto the flap, and its rearward movement along the flap, for $3° \leq \sigma_f \leq 10°$, caused the greatly increased slope of the hinge-moment curve For values of $\sigma_f \gtrsim 17°$, the shock wave was essentially fixed at the trailing edge

The contribution to the hinge-moment coefficient due to flap oscillations, $C_{h_{\dot\sigma_f}}$, were evaluated from the periodic response to small-amplitude harmonic flap oscillations The flap motion was specified as

$$\sigma_f(t) = \sigma_{f_m} + \sigma_{f_1} \sin \omega t \tag{14}$$

where σ_{f_m} ranged from 0° to 20°, and σ_{f_1} was small, usually 0 5°. The reduced frequency of the motion, $k = \omega l/V$, was held fixed at 0 15 for all cases Starting from an initial steady solution obtained at $\sigma_f = \sigma_{f_m}$, computations were carried out for three cycles of the motion to ensure that a periodic solution had been obtained The hinge-moment damping coefficient was evaluated from the component of the aerodynamic response that was 90° out of phase with the flap motion The results are shown in Fig 13 as a function of the mean flap deflection angle Note that at the transonic flow conditions considered in Ref 10, the damping coefficient is a highly nonlinear function of the flap deflection For values of $\sigma_{f_m} \leq 3°$ the coefficient is negative, or dynamically stabilizing However, for mean flap deflections ranging between 3° and 17°, the coefficient is positive (dynamically destabilizing) and would cause an unconstrained flap oscillation to increase in amplitude The nonlinear behavior of the damping coefficient is qualitatively related to the behavior of the static hinge moment In particular, the decrease in dynamic stability is linked to the increase in slope of the static hinge-moment curve (see Ref 22 for a detailed discussion)

4 3 Flap Motion Histories

Oscillatory time-histories of the flap motion were generated using the flap equation of motion, Eq (13), with the instantaneous hinge-moment coefficient specified by the nonlinear aerodynamic model In these computations, the flap moment of inertia was chosen to give a value of reduced frequency close to the one specified for the characteristic-motion computations After initial values of the flap deflection angle and velocity were specified, the equation of motion was solved numerically to obtain the motion histories At each time step the hinge-moment coefficient was specified by Eq (4), where the terms $C_h(\sigma_f(t))$ and $C_{h_{\dot\sigma_f}}(\sigma_f(t))$ were obtained from table look-ups in Figs 12 and 13, respectively Corresponding histories of the flap motion were also predicted using the coupled-equations approach

Time-histories of flap motion, generated with both the modeling approach and the coupled-equations approach are shown in Figs 14 and 15 The motion resulting when the flap was released from rest with an initial deflection angle $\sigma_{f_0} = 4\,0°$ is shown in Fig 14 In this case the stabilizing portion of the hinge-moment damping curve governed the motion, and the amplitude of the oscillation decayed smoothly In contrast, when the flap was released with a slightly larger initial deflection, $\sigma_{f_0} = 4\,5°$, the dynamically destabilizing portion of the damping curve caused the amplitude of the oscillation to grow rapidly (Fig 15) In both cases, however, the motion histories obtained using the aerodynamic mathematical model were in good agreement with the "exact" motion histories obtained from the coupled equations. This confirmed the ability of the mathematical model, Eq (4), to describe the unsteady aerodynamic response in this highly nonlinear transonic flow On the other hand, a motion history generated with an aerodynamic model which does not account for nonlinear unsteady aerodynamic contributions (i e., Eq (4), but with the value of the damping term $C_{h_{\dot\sigma_f}}$ held fixed for all σ_f at the value obtained at $\sigma_f = 0°$) failed to predict the undamped growth of the of the flap oscillation (Fig 15)

5 DISCUSSION

The agreement between the force and moment histories obtained for the delta wing using the modeling approach and those obtained from direct CFD computations confirmed the ability of the model to describe the unsteady aerodynamic response to complex motions, at least within the limited range of motions and rates considered Similarly, the agreement between the flap motion histories obtained with the coupled-equations approach and those generated with the model confirmed the capacity of the model to adequately describe the unsteady, nonlinear aerodynamic response to arbitrary flap motions The success of the modeling approach in this case points the way to approach future problems involving unsteady motions In our view, for cases where mathematical models are valid, the modeling approach will be more economical and more versatile than the coupled-equations approach Further, use

of the modeling approach will give better insight into the underlying physics than can be obtained through use of the coupled-equations approach

First, regarding costs, we note that the computation of motion histories from the vehicle equations of motion requires negligible computational effort in comparison with that needed for computations of time-dependent flowfields Consequently, with the modeling approach, once the initial effort of evaluating the aerodynamic responses to the characteristic motions is expended, computation of motion histories would be relatively inexpensive Regarding versatility, the modeling approach makes it easy to introduce changes into the aircraft's equations of motion (e g . changes in vehicle mass or moment of inertia, or inclusion of a model of a control system) and to evaluate their effects at low cost, since the aerodynamic data within the mathematical model will remain unchanged In contrast, in the coupled-equations approach the simplest change in the aircraft's equations of motion would require a complete reevaluation of the flowfield and motion response

Second, the modeling approach would appear to give better insight into the physics governing the unsteady flow than would the coupled- equations approach If an undamped or divergent motion results from coupled-equations computations, it would be difficult to identify the aerodynamic phenomena causing the instability On the other hand, computations carried out in terms of the characteristic motions permit an investigation of the underlying aerodynamic mechanisms For the transonic flap motions considered in Ref 10, these computations indicated that it was the rearward movement of the upper-surface shock wave that caused the large change on the slope of the static hinge-moment curve It was also possible to show how the change in the static hinge-moment coefficient was related to the destabilizing behavior of the hinge-moment damping coefficient Further, the modeling approach is compatible with established methods for determining the stability of motions In the case of the flap, knowledge of the behavior of the damping coefficient with increasing deflection, obtained from the characteristic motion computations, permitted prediction of the change from a damped to an undamped oscillation with increased initial deflection angle that was subsequently observed Thus, the nonlinear modeling approach would appear to be the method of choice in the design of flight control systems and in flight simulations

6 CONCLUDING REMARKS

Applications of CFD methods to demonstrate the regimes of validity of nonlinear aerodynamic mathematical models for the case of a maneuvering delta wing, and for the case of an oscillating flap on an airfoil in transonic flow, are discussed The assumptions underlying the development of the models are reviewed, and the ensuing limitations on the aerodynamic phenomena that the resulting models can accommodate are discussed The class of mathematical models considered, where the nonlinear aerodynamic responses are continuous single-valued functions of the motion variables, was found to be adequate for the types of aerodynamic phenomena that were in play Extension of the mathematical models to encompass a wider variety of possible aerodynamic phenomena, including bi-valued aerodynamic responses and time-varying responses to steady motions is currently being pursued

The economic and conceptual advantages of the modeling approach over that of coupled, simultaneous, solutions of the flowfield and kinematic equations of motion in predicting flight vehicle motion histories are illustrated The modeling approach, when valid, eliminates the need for costly repetitive computation of flowfield solutions when multiple motion histories must be determined

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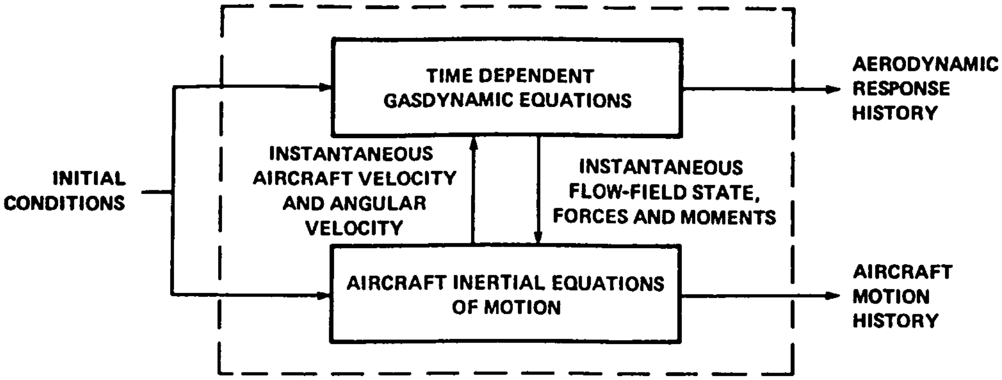
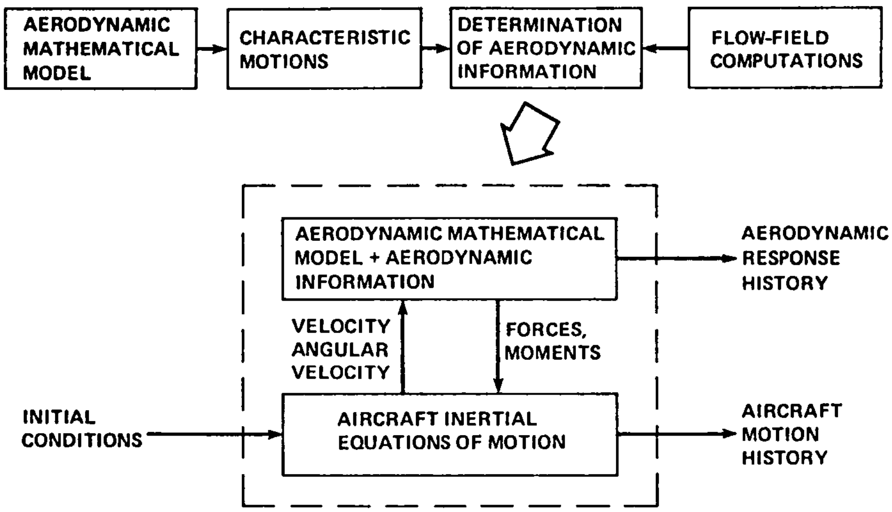


(a) Coupled flowfield and inertial equations approach.



(b) Aerodynamic mathematical modeling approach.

Fig. 1. Approaches to applying CFD methods to predict aerodynamic responses and aircraft motion histories.

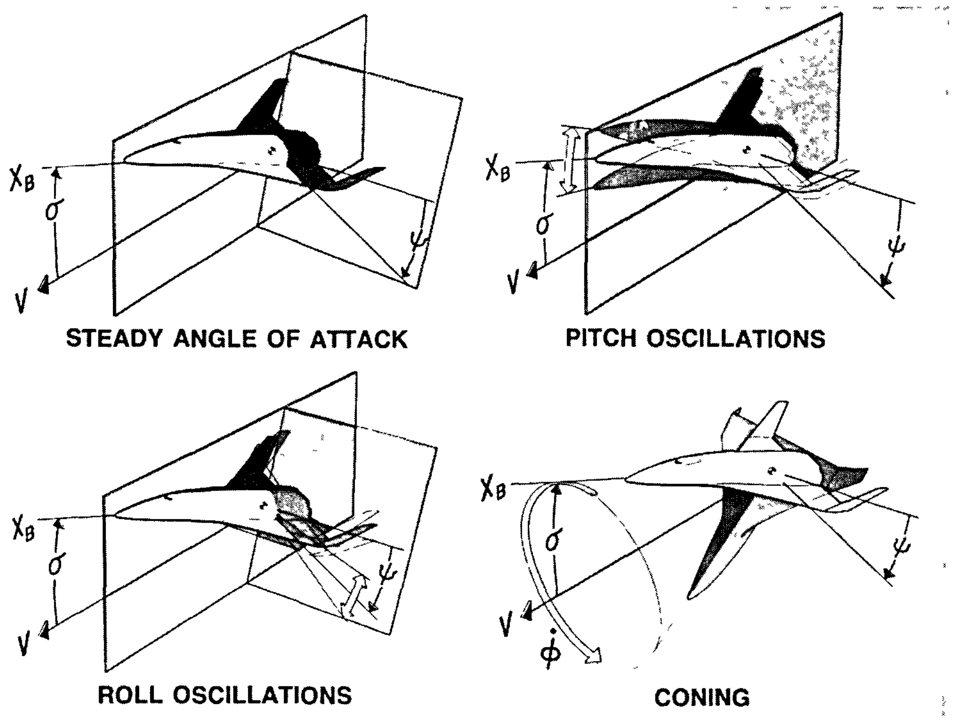


Fig 2. Aerodynamic axis system and characteristic motions obtained assuming linear variation of the response on the motion rates

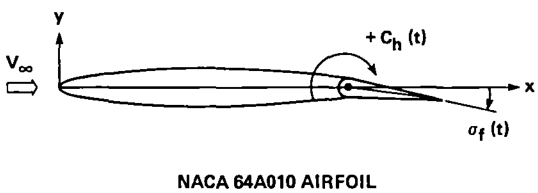


Fig. 3 Coordinates and notation for flap motion.

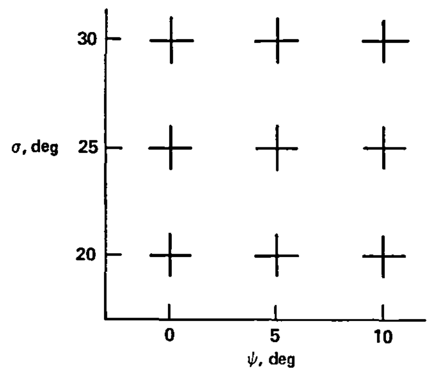


Fig. 4 Range of angle of attack σ and roll angle ψ considered in determining aerodynamic coefficients for the mathematical model (Ref. 13)

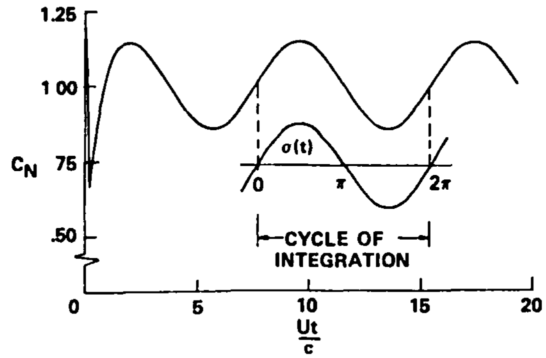


Fig. 5. Fourier integration of the normal-force response to evaluate the aerodynamic coefficients from Equations (7)-(9) (Ref. 13).

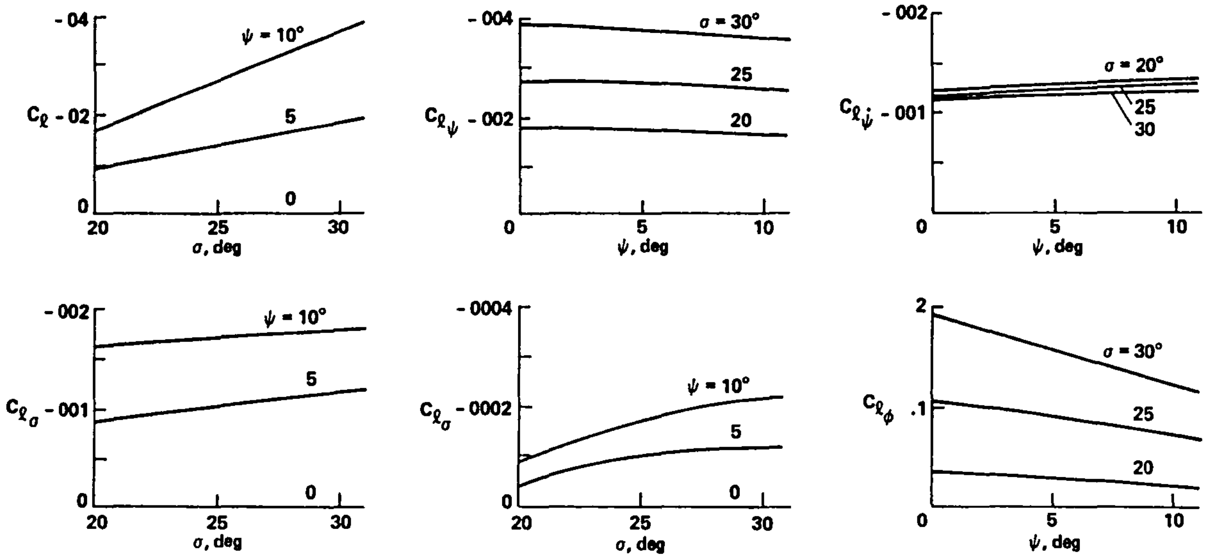


Fig. 6. Rolling-moment coefficients of delta wing evaluated from characteristic motions (Ref. 13).

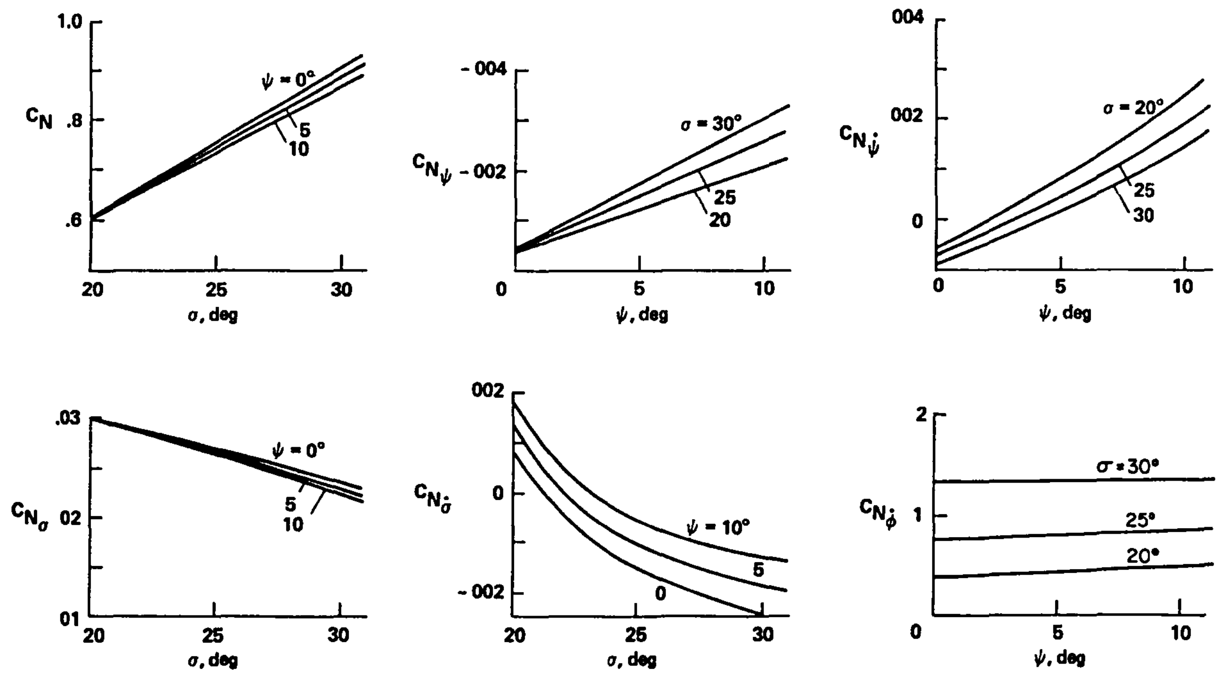


Fig. 7. Normal-force coefficients of delta wing evaluated from characteristic motions (Ref. 13).

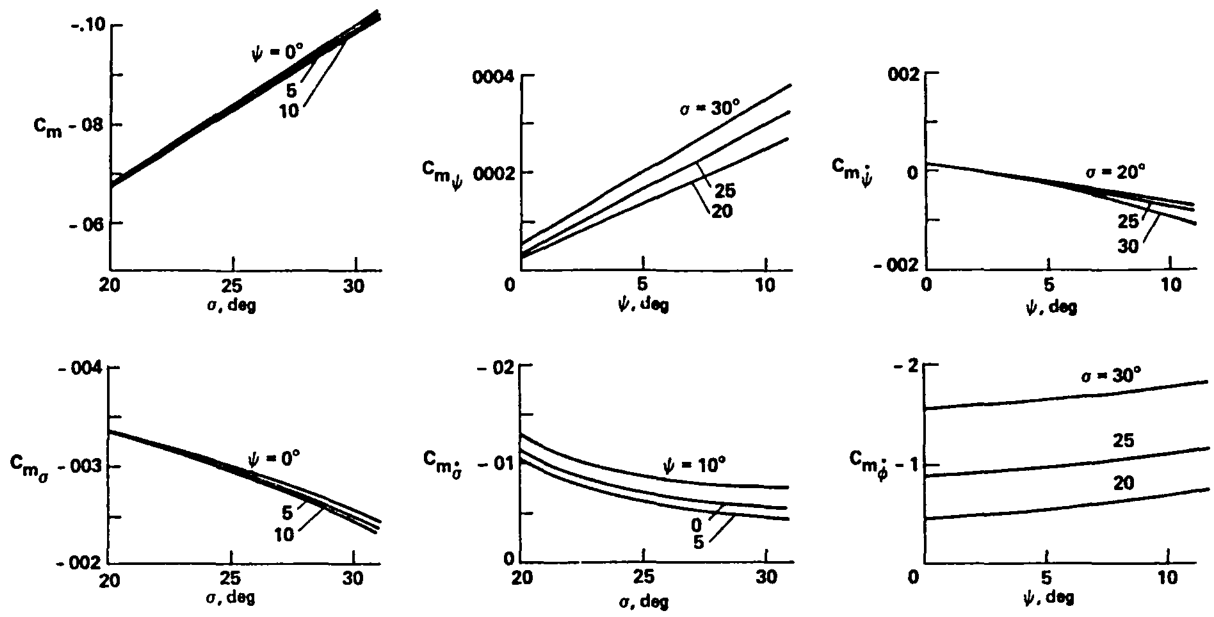


Fig. 8. Pitching-moment coefficients of delta wing evaluated from characteristic motions (Ref. 13).

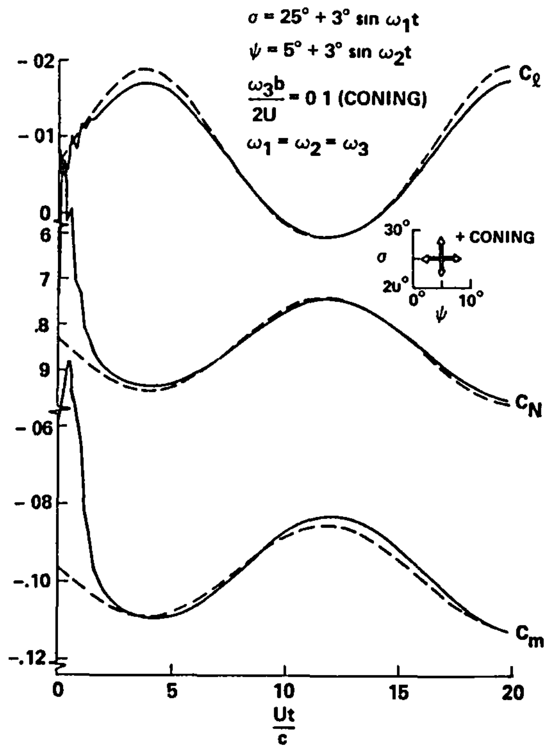


Fig. 9. Aerodynamic response of delta wing to combined pitch oscillations, roll oscillations, and coning motion (Ref. 13).

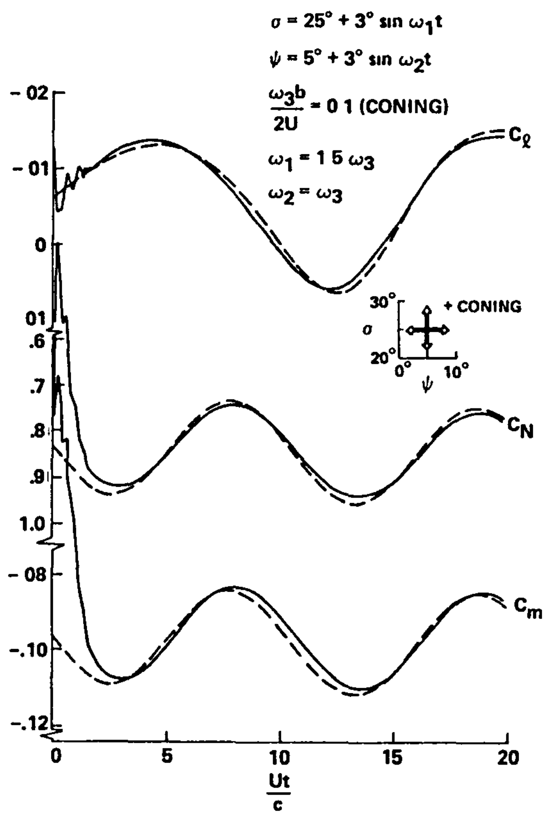


Fig. 10. Aerodynamic response of delta wing to combined pitch oscillations, roll oscillations, and coning motion (Ref. 13).

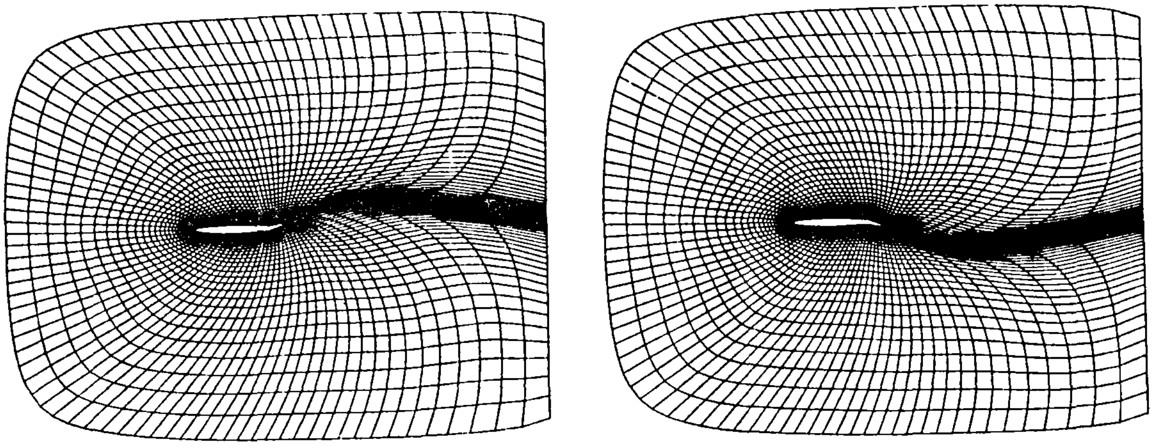


Fig. 11. Typical body-conforming grids with flap at extreme deflection angles (Ref. 10).

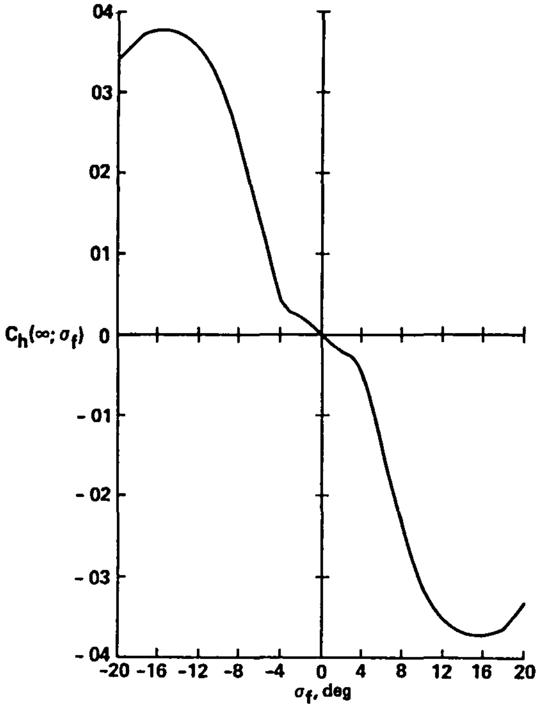


Fig. 12. Static hinge-moment coefficient: M = 0.8 (Ref. 10).

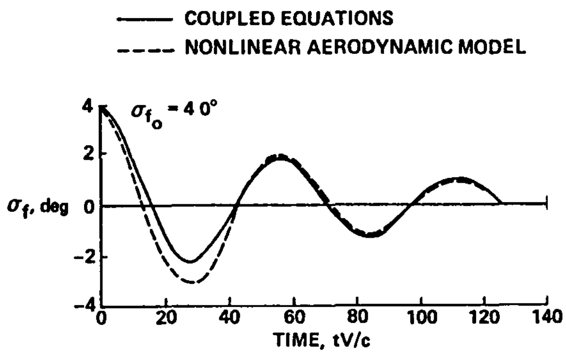


Fig. 14. Time-history of flap deflection; $\sigma_{f_o} = 4.0°$ (Ref. 10).

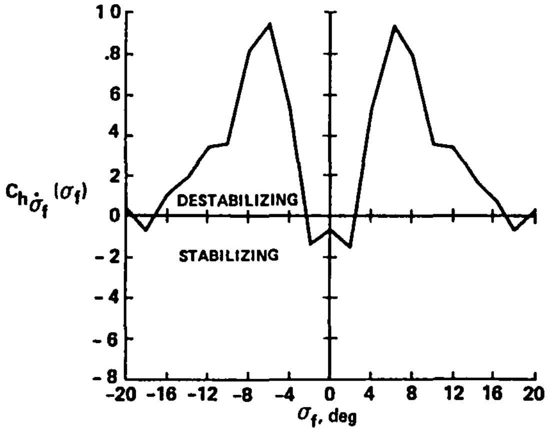


Fig. 13. Hinge-moment damping coefficient: M = 0.8 (Ref. 10).

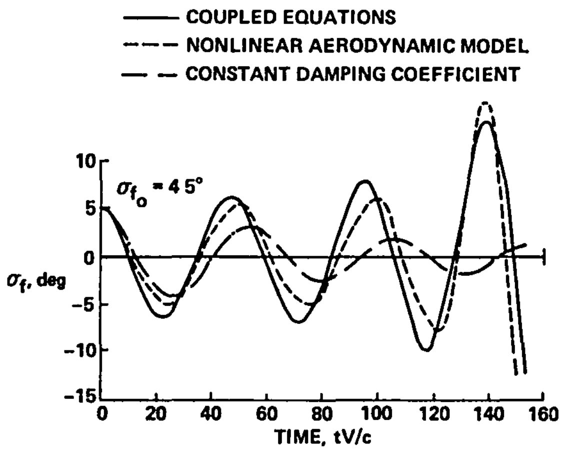


Fig. 15. Time-history of flap deflection, $\sigma_{f_o} = 4.5°$ (Ref. 10).

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15 Supplementary Notes

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16 Abstract

Applications of CFD methods to determine the regimes of applicability of nonlinear models describing the unsteady aerodynamic responses to aircraft flight motions are described. The potential advantages of computational methods over experimental methods are discussed and the concepts underlying mathematical modeling are reviewed. The economic and conceptual advantages of the modeling procedure over coupled, simultaneous solutions of the gasdynamic equations and the vehicle's kinematic equations of motion are discussed. The modeling approach, when valid, eliminates the need for costly repetitive computation of flow field solutions. For the test cases considered, the aerodynamic modeling approach is shown to be valid.

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